AEDC-TR-55-6

TEST FACILITIES FOR ULTRA-HIGH-SPEED AERODYNAMICS

By

R. Smelt; GDF, ARO, Inc.

June 1955

ARNOLD ENGINEERING DEVELOPMENT CENTER

AIR RESEARCH AND DEVELOPMENT COMMAND

US AF

AEDC-TR-55-6

TEST FACILITIES FOR ULTRA-HIGH-SPEED AERODYNAMICS

By

R. Smelt; GDF, ARO, Inc.

June 1955

Contract No. AF 18(600)-1233

CONTENTS

ILLUSTRATIONS

SUMMARY

Hypersonic wind tunnels with test section temperatures approaching liquefaction are shown to have an upper limit in Mach number of about 12 with Reynolds numbers in the gas dynamic flow regime. This limit arises both from the problem of cooling the tunnel walls, and from the requirement for correct simulation of air temperature above Mach number 12, to reproduce dissociation and ionization phenomena.

Three types of facilities for higher Mach number, providing correct flight temperatures, are discussed:

1. A conventional wind tunnel with sufficiently high supply temperature. This appears only practicable at Reynolds numbers in the slip-flow regime.

2. An intermittent facility operating for a few milliseconds, using shock tube techniques. Maximum Reynolds numbers are about 100 times greater than in the preceding type.

3. A supersonic tunnel in which the air is heated after reaching high supersonic speed. Techniques for achieving this are discussed.

NOMENCLATURE

A	Cross-sectional area
A^*	Throat area
A_t	Cross-sectional area of the test section
c	Specific heat of the wall material
c_p	Specific heat, constant pressure
H	Heat transfer per unit area of wind tunnel wall per second
k	Thermal conductivity of the wall material
k_h	Heat transfer coefficient
M	Mach number
p_o	Supply pressure
T	Temperature
T_m	Liquefaction temperature of the wall material
T_o	Supply temperature
T_r	Recovery temperature
T_w	Wall temperature
t	Time
R	Reynolds number
r	Test-section Reynolds number per foot of model length
v	Velocity
γ	Specific heat ratio
μ	Coefficient of viscosity at the test section
ρ	Density

I. INTRODUCTION

It is well known that ordinary supersonic wind tunnels, operating from an air supply at approximately room temperature, have an upper limit in Mach number which is determined by the commencement of liquefaction of the air around the model in the test section. For a wind tunnel with a supply pressure of one atmosphere and a supply temperature of 80°F, the air in the test section reaches the liquefaction point at a Mach number of 5. This does not necessarily imply that the wind tunnel ceases to give useful results exactly at a Mach number of 5; the exact limit is dependent upon the magnitude of the local additional expansion around the test object, and the extent to which temperatures lower than the liquefaction temperature are permissible, either because of local supersaturation during the extremely short time at lower temperature, or because the heat release is not significant when the degree of liquefaction is small. These factors, primarily dependent upon the type and size of model and the required precision of measurement, do not change the limit Mach number greatly from the value deduced directly from "equilibrium" liquefaction data.

In the hypersonic tunnels now operating in a few laboratories throughout the country (Refs. 1, 2, 3), the onset of liquefaction has been delayed by heating the supply air. By this means, an extension of the maximum Mach number to about 10 or 11 has been obtained at the expense of a considerable increase in the complexity of the wind tunnel. Further extension of the Mach number range of hypersonic wind tunnels by means of heating is possible, but there are practical engineering limits to this process. Furthermore, there is also a physical limit to the simulation of flight characteristics at high Mach number in wind tunnels of this type.

In the following paragraphs these engineering and physical limitations which prevent further increase in Mach number of hypersonic wind tunnels are discussed in a rough quantitative manner. Extension of the Mach number range of test facilities beyond these limits calls for the development of novel types of equipment, capable of handling temperatures and pressures well beyond current wind-tunnel experience. Some of the proposals which have been made for test facilities of this type are discussed later in the report.

II. ENGINEERING LIMITATIONS TO HYPERSONIC WIND TUNNELS

As stated above, the temperature required to avoid significant effects of liquefaction in a hypersonic wind tunnel is dependent upon the intended application of the wind tunnel, i.e., upon the type of model being tested and the required precision of the test results. A fairly representative picture is obtained by assuming that the air at ambient pressure in the test area should be at a temperature equal to the equilibrium dewpoint. This implies that the supply temperature T_o should increase with increasing Mach number M, and also with increasing supply pressure p_o, in accordance with the following relation:*

$$\frac{667}{T_o}\left(1 + \frac{M^2}{5}\right) = 4.7 + 3.5 \log_{10}\left(1 + \frac{M^2}{5}\right) - \log_{10} p_o \qquad (1)$$

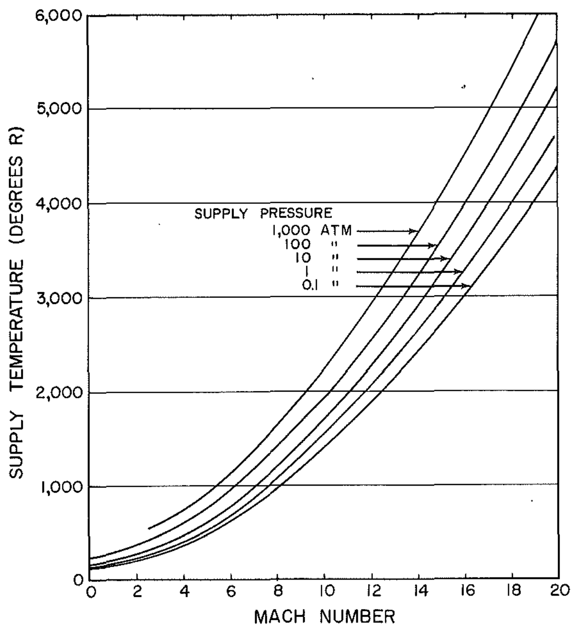

Fig. 1. Approximate Temperature Required to Avoid Liquefaction of Air in Hypersonic Wind Tunnels

The relation of supply temperature with Mach number and supply pressure is shown graphically in Fig. 1. The rapidity of the rise in supply temperature with increasing Mach number shown in the figure makes it evident that the upper limit of Mach number will be dependent upon the engineering ability to handle high temperature gases, i. e., upon the effectiveness of the cooling arrangements which can be made in wind

*This expression combines an approximate Clausius-Clapyron relation between liquefaction temperature and pressure, based on the experiments of Dodge and Dunbar (Ref. 4), with expansion characteristics for a perfect gas with the specific-heat ratio $\gamma = 1.4$. This second assumption is of course increasingly inaccurate with increasing supply temperature.

tunnels of this type.

There is no great engineering difficulty in obtaining the required supply temperature. When the required temperature exceeds the maximum which can be achieved by conventional convective heating, additional temperature rise can be obtained by such methods as adiabatic compression or arc heating. There is also no serious engineering difficulty in obtaining the required supply pressures for hypersonic wind tunnels; commercial equipment and techniques are available for pressures ten times greater than the maximum now being employed in such tunnels. The engineering limit is defined by the problem of cooling the walls of the wind tunnel exposed to high-velocity air streams at these high temperatures and pressures.

To examine this upper limit more closely, it is necessary to obtain a rough measure of the cooling requirements as a function of Mach number. The heat transfer H per unit area of the wind tunnel wall per second may be written in the form:

$$H = k_h \; c_p \; \rho v \; (T_r - T_w) \tag{2}$$

In this expression the quantities k_h and T_r vary somewhat with the local Mach number and Reynolds number of the air, i.e., with position along the tunnel; but their variation is not significant compared with the changes in ρ and v. In practice, therefore, the cooling problem in a hypersonic wind tunnel is at its worst in areas near which the quantity $\rho \cdot v$ is a maximum, i.e., near the throat of the wind-tunnel nozzle. The heat-transfer rate is in fact roughly inversely proportional to the cross-sectional area of the wind tunnel at every point.

The magnitude of the cooling problem is evidently increased with increasing supply pressure, and there would in fact be no difficulty in operating a tunnel at extremely high Mach number if it were permissible to use a sufficiently low supply pressure. There are several reasons why this is not possible. One obvious problem would be to observe or measure aerodynamic characteristics in the test area at extremely low density.

A further requirement for high supply pressure is inherent in the aerodynamic applications of hypersonic wind tunnels. Very few problems at hypersonic speeds are associated with perfect-gas aerodynamics; the most important questions relate to the rate of heating or the drag of a hypersonic vehicle, and these require close simulation of boundary-layer characteristics on a test model. In other words, it will always be important in a hypersonic tunnel to obtain Reynolds numbers as close to flight values as possible. In practice, of course, as in lower-speed wind tunnels, exact simulation of flight Reynolds numbers will frequently be impracticable, and the requirement will be reduced to a need for simulation of the general character of the boundary-layer flow. As examples, a turbulent boundary layer in a flight should of course be simulated by a turbulent layer on the test object, and flight in the gas dynamics region will not be adequately represented if wind tunnel Reynolds numbers are in the slip-flow region (defined as $R < 10^4 M^2$).

An approximate assessment of the cooling problem as influenced by test Reynolds number can be made very simply. If the test-section Reynolds number per foot of model length is denoted by r, then:

$$r = \rho v/\mu \quad \text{at test section.}$$

From Equation (2) above:

$$H = k_h \; c_p \; r\mu \; \frac{A_t}{A} \; (T_r - T_w) \tag{3}$$

This equation has been employed to construct Fig. 2, which shows the variation in maximum heat-transfer rate at the throat with Mach number and Reynolds number per foot, assuming that T_r is equal to the supply temperature as shown in Fig. 1, and that an acceptable wall temperature at the throat is 1000° R (540° F). There is considerable uncertainty in assessing a value of k_h appropriate to boundary-layer conditions at the throat, in view of the high pressure gradient in this region, the undefined Reynolds number, and the lack of knowledge of transition phenomena in such a region. Based on hypersonic tunnel design experience, a constant value of 0.0014 has been chosen; this is equivalent to an assumption that transition to turbulent flow takes place in the vicinity of the throat, and is probably somewhat conservative. A more exact value could be obtained by making a step-by-step calculation of the growth of the boundary layer from the subsonic section of the nozzle through the throat; a considerable amount of analysis of this type, as yet unpublished, has been made by Professor Paul A. Libby of Polytechnic Institute of Brooklyn. For the very general purpose of the present survey, the simpler approach of assuming a constant heat-transfer coefficient is adequate.

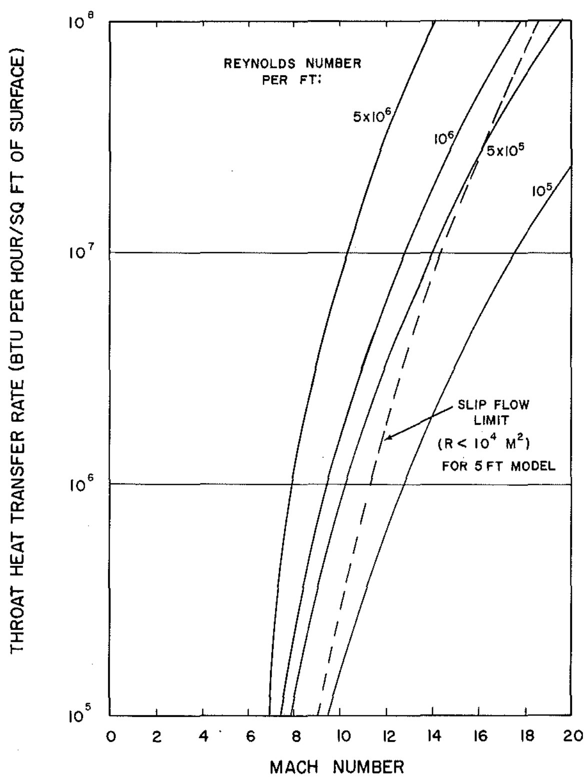

Fig. 2. Maximum Heat-Transfer Rate at Hypersonic Tunnel Throat; Variation with Mach Number and Reynolds Number

The throat heat-transfer rate shown in Fig. 2 is given over a range of Reynolds number from 5 millions to 0.1 million per foot, which covers the range of typical hypersonic tunnels. The Reynolds number boundary below which slip-flow phenomena begin to appear is shown in the figure for a model length of 5 ft.; equivalent curves for other lengths are easily determined.

The heat-transfer rate has been expressed in Btu per hour per square foot of surface, since this is the form most familiar to heat engineers. For comparison, typical heat-transfer rates in high-capacity, high-temperature steam boilers are in the range from 45,000 to 80,000 Btu/hr/sq ft,* below the lower border of Fig. 2.

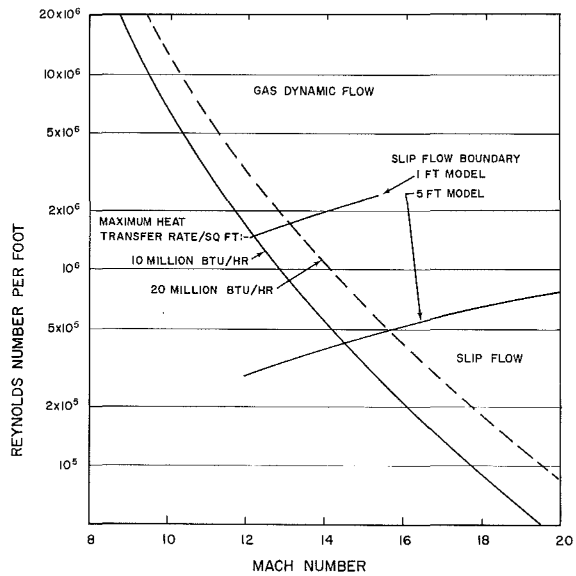

Fig. 3. Limiting Mach Number-Reynolds Number Relation in Continuous Hypersonic Tunnels, from Cooling Considerations

The highest heat-transfer rates in current engineering practice, to the knowledge of the author, are encountered in rocket motors; and the cooling problem at the throat of such motors is very similar to the hypersonic tunnel problem. It is reasonable to assume therefore that if all the rocket developments in liquid jacket cooling, boiling fluid cooling, or film cooling are adopted for use in hypersonic tunnels, similar maximum heat-flow rates, of the order of 10^7 Btu/hr/sq ft, should be possible. The corresponding maximum Mach number, as a function of Reynolds number per foot, is shown as the full curve in Fig. 3. This figure also includes a similar curve, shown dotted, illustrating the effect of doubling the maximum heat-transfer rate. It is evident that such a development, if possible, would only permit the limit Mach number to be increased by about 1.0.

This assumption that heat-transfer rates equal to those of rocket motors should be attainable in hypersonic tunnel practice is an optimistic one, since the hypersonic tunnel throat presents an additional problem, in accuracy of profile,

*From Kent's Mechanical Engineers' Handbook, Power, 12th Edition, Section 7, p. 16.

not present in rocket motors. To illustrate this problem, consider a hypersonic tunnel with a test section 1-ft square, utilizing a two-dimensional nozzle to obtain a Mach number of 10. The ordinate at the throat is then only 0.022 in., and a change of only 0.001 in. in this ordinate will change the test-section Mach number by 0.1 and the test-section static pressure by 7 percent. An identical change across the whole width of this throat can of course be corrected; but if the large heat flow produces uneven temperatures and distortion in the throat wall, the resulting transverse nonuniformities in the flow cannot be eliminated.

It is evident that a hypersonic tunnel aimed at reaching limit heat-transfer conditions at the throat should preferably have an axially symmetrical nozzle, not only to reduce distortion but also to minimize the surface area in the vicinity of the throat. The axially symmetrical nozzle has not been adopted generally in hypersonic wind tunnels however, for two reasons: (1) a fear that the axial pressure distribution may be nonuniform because of focussing of wall disturbances, and (2) a requirement for variation in Mach number of the nozzle, which is more difficult to achieve in an axially symmetrical design.

III. PHYSICAL LIMITATIONS TO HYPERSONIC WIND TUNNELS

Hypersonic wind tunnels of the type considered in the last paragraph do not simulate the temperatures which exist around an object in flight, but operate at ambient temperatures close to the liquefaction temperature of air. If air were a perfect gas with constant specific heats, over the whole temperature range experienced in tunnel and flight, all temperatures would be proportional, and an accurate picture of flight temperature distributions would be obtained from tunnel tests provided that surface temperatures were simulated. Unfortunately air is not a perfect gas, and its deviations become significant at the high temperatures generated in very high speed flight.

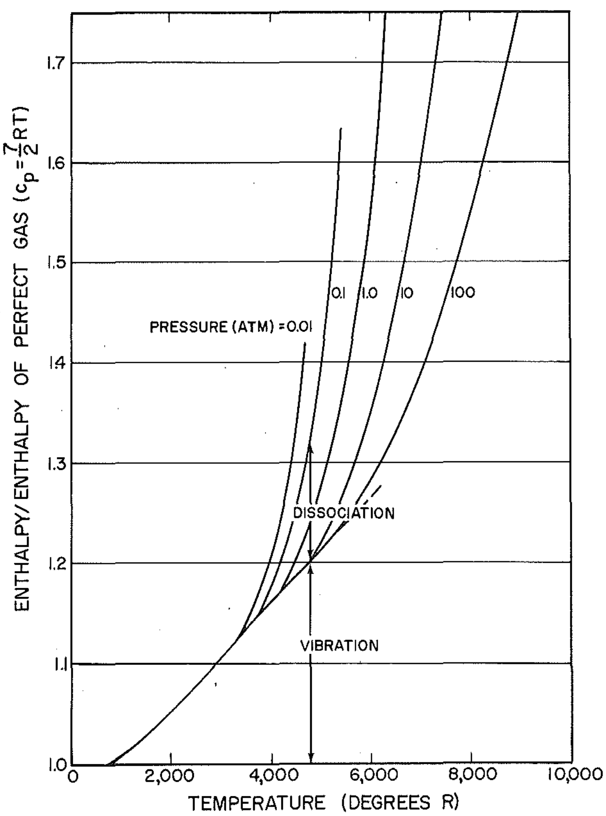

Fig. 4. Enthalpy of Air at High Temperatures

These deviations are presented in Fig. 4, in which the enthalpy at high temperatures is compared with the value which would be expected if the specific heat remained constant at the low temperature value. This figure has been constructed from the tables of Ref. 5. It shows how the enthalpy increases with increasing temperature, as the additional energy is first absorbed in exciting the vibrational degree of freedom of the molecule, and then in dissociation of the molecule. The absorption of energy in vibration is not greatly affected by pressure, but dissociation proceeds much more rapidly at lower pressures.

In the flow around a model in a hypersonic wind tunnel, the air is at low temperature and behaves essentially as a perfect gas, with enthalpy corresponding to the value of 1.0 on Fig. 4. Figure 4 thus expresses the ratio between the actual enthalpy in flight and the value simulated in the wind tunnel, as a function of flight temperature. The exact extent to which this difference changes the aerodynamic parameters of course depends upon the details of the flow; at present comparisons can only be made in a few cases where the flow of the real gas with vibration and dissociation has been computed theoretically. The characteristics of a normal shock in a real gas have been calculated by Bethe and Teller (Ref. 6) and later calculations are also given in Ref. 5. The laminar flow in the boundary layer has been computed by Moore (Ref. 7), Crown (Ref. 8) and others. Generalizing these calculations, it appears that changes in the temperature and density of the same order as the change in enthalpy shown in Fig. 4 may be expected, although the pressures are not generally modified to the same extent.

With this background, Fig. 4 can be used to estimate roughly the maximum Mach number at which hypersonic wind-tunnel test results can be applied to flight conditions. There is evidently a Mach number range over which the changes in air properties in flight are so small that they can be neglected completely. The point where the enthalpy has changed by one percent has been arbitrarily selected as defining the upper limit in this range. There is a second regime in which significant but small differences between wind tunnel and flight characteristics are to be expected; these differences might be treated as small corrections to the wind-tunnel test results, the corrections being based largely upon theoretical treatment of the consequences of air imperfection. From the point of view of simplification of the theoretical treatment, it appears advantageous to define the upper limit of this regime as the point at which dissociation becomes significant, thereby confining the corrections to vibrational effects only. In the third and highest range of Mach number, vibrational effects are large and appreciable dissociation is also present. Theoretical treatment is then more complex, the differences between tunnel and flight characteristics are large, and it appears that the hypersonic wind tunnel has lost much of its utility.

The temperatures defining the limits of these regimes can be obtained directly from Fig. 4. At first sight, the limit Mach number could be obtained by equating these temperatures to the stagnation or recovery temperature in flight, since these temperatures are attained behind a normal shock or at the inner edge of the boundary layer on an insulated wall. From the practical point of view, this would be over-conservative; a vehicle flying at very high Mach numbers is not likely to have any extensive areas of stagnation conditions, and its wall temperature must be well below the recovery temperature. Under these conditions, the maximum temperature occurs in the boundary layer away from the wall; the temperature rise is about one-fourth of the full stagnation value (Ref. 9). The Mach number limits for the three regimes under these conditions are given in Table 1 below. In preparing this table the dissociation curve of Fig. 4 for a pressure of 0.01 atmosphere has been used, since this corresponds approximately to boundary layer conditions at the top of the stratosphere.

Table I. Ranges of Application of Hypersonic Wind Tunnels

	Range I Results Applicable to Flight without Correction	Range II Corrections for Vibration Effects Required	Range III Large Differences between Tunnels and Flight
Maximum local temperature (^{o}R)	Up to 1000	1000 - 3400	Above 3400
Corresponding Stagnation temperature (^{o}R)	Up to 2800	2800 - 12,400	Above 12,400
Flight Mach number in stratosphere ($T = 400^{o}R$)	Up to 5.5	5.5 - 12	Above 12

From this table it is evident that the hypersonic tunnel loses much of its value at the upper limit of Range II, i. e., at a Mach number of about 12. It is interesting to compare this conclusion with the curve of Fig. 3, which shows that the practical cooling problem at the tunnel throat brings the Reynolds number almost down to slip-flow values at a Mach number of 12.

It will be observed from Table I that corrections for the vibrational effects in flight are required over quite a wide range of hypersonic tunnel Mach number, i. e., above a Mach number of 5.5. There is at present very little experimental or theoretical data upon which to base corrections of this type. Probably the greatest need at present is for an adequate treatment of the turbulent boundary layer in a real gas, to permit the interpretation of hypersonic tunnel measurements of heat-transfer rate and skin friction. The problem is complicated by the relaxation time of the vibrational degree of freedom, but even an equilibrium theory would be of considerable assistance. The new facility at the Freeport Laboratory of the Polytechnic Institute of Brooklyn, with its ability to duplicate flight temperatures at moderately high Mach numbers, should give much-needed information in this area.

IV. FACILITIES FOR MACH NUMBERS ABOVE 12

In the highest range of Table 1, above a Mach number of 12, it appears necessary that the test conditions should simulate the actual temperature of the air in flight, since the changes in air properties at high temperature will have a predominant effect upon the whole aerodynamic picture.

To be strictly correct, it also appears necessary to have correct simulation of pressures around the test object, since the degree of dissociation is changed quite appreciably by change in pressure. It is evident from Fig. 4 however that the effect of a reduced test pressure could be compensated approximately by a reduction in the stagnation temperature, at least over a moderate range in pressure.

The influence of pressure on the relaxation effects associated with dissociation and vibration presents more difficult problems. If these effects are important in flight, their simulation on the model requires an equal number of molecular collisions in a comparable length of test object, and this resolves itself into a requirement for equal Reynolds numbers in model and flight. Of course, if the flight Reynolds number is so high that relaxation effects are negligible, the requirement for equal model Reynolds number is replaced by a requirement that the model Reynolds number should be high enough to avoid significant relaxation distances in this case also.

It is evident from the earlier examination of the cooling problems of hypersonic wind tunnels (Section II) that Mach numbers above 12 will not be attainable except at very low Reynolds numbers. The requirement for correct test-section temperature of course considerably enlarges the cooling problem; the test-section temperature is increased by a factor of about 4 compared with that of an ordinary hypersonic tunnel, so that the heat transfer at all points is multiplied by about the same factor from this cause alone. There are, however, two other strong effects which further increase the heating rate at the throat of such a tunnel:

1. The increased test-section temperature increases the viscosity, and reduces the Reynolds number. To obtain the same Reynolds number per foot as in the hypersonic tunnel considered in earlier sections, it is necessary to multiply the density by a factor of almost 4. This of course increases the heat-transfer rate at the throat by the same factor.

2. Throughout most of the expansion process in the nozzle of a temperature-simulating tunnel, the vibrational mode of the air is almost fully excited, and there is appreciable dissociation. Under these conditions the specific heat is greatly increased and the specific heat ratio γ has fallen to a value between 1.2 and 1.3. This considerably increases the ratio of test-section area to throat area for a given Mach number; an approximate estimate of this effect has been made, and is shown in Fig. 5. Both this ratio and the specific heat enter into the expression (Equation 3) for the heat transfer rate at the throat, with the result that very large increases in this heat rate can be expected when the imperfections of the air are taken into account.

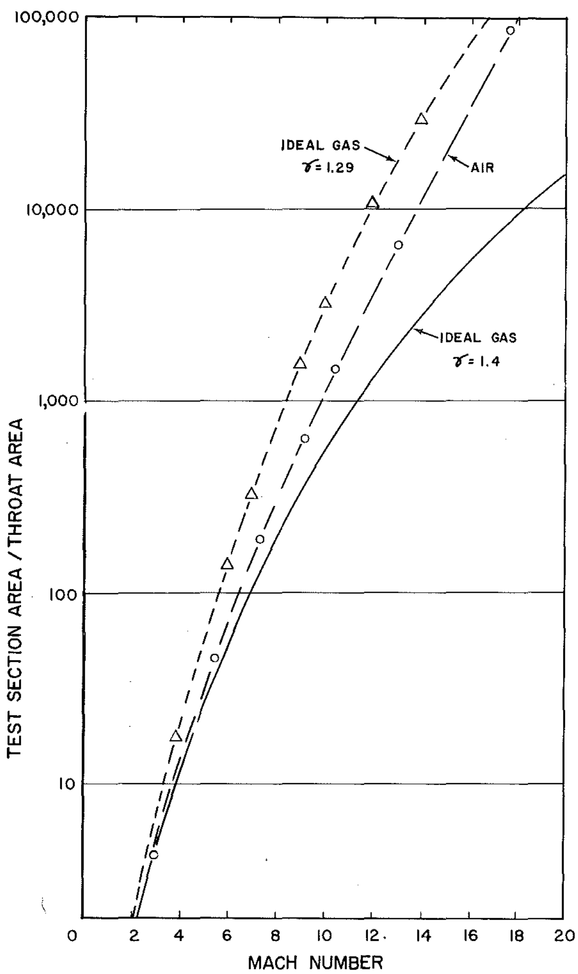

Fig. 5. Test Area/Throat Area Ratio in Temperature-Simulating Hypersonic Tunnels

The curves shown in Figs. 6 and 7 have been constructed to provide a rough illustration of the throat-cooling problem when all these factors are taken into account. They should be contrasted with the similar curves shown in Figs. 2 and 3 for the heat-transfer conditions in a conventional hypersonic tunnel. It should be emphasized that the heat-transfer phenomena at the throat are so complex under the high temperature conditions now being considered, and the data on

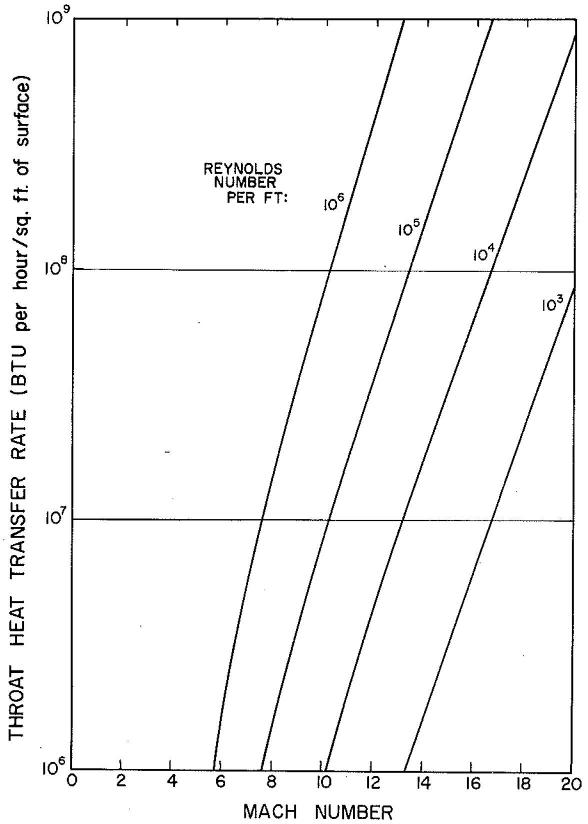

Fig. 6. Maximum Heat-Transfer Rate at Throat of Temperature-Simulating Tunnel; Variation with Mach Number and Reynolds Number

air properties are so sparse, that Fig. 6 must be regarded only as an order-of-magnitude estimate of the probable heat-transfer rate and is probably an under estimate of the total heat flow in this area, since important effects such as radiation have been completely neglected. It serves the purpose however of demonstrating that a wind tunnel simulating correct flight temperatures and Reynolds numbers in the gas dynamic regime is quite impractical. It appears possible to make such a wind tunnel to give low slip-flow Reynolds numbers at Mach numbers around 20, and to extend out of the slip-flow region at the lower Mach numbers around 10. While a facility of this type would clearly have some utility, the practical designer of vehicles for flight at these extreme Mach numbers will undoubtedly be more concerned with aerodynamic problems at the higher Reynolds numbers which correspond to flight conditions at lower altitudes. Figures 6 and 7 invite speculation on the extent to which it may be possible in the future to improve cooling conditions at the throat of a high-temperature tunnel by development of new cooling techniques. There is no doubt that some improvement will occur; in the opinion of the author, the full gain to be offered by the use of cold air injection ahead of the throat has not been realized up to the

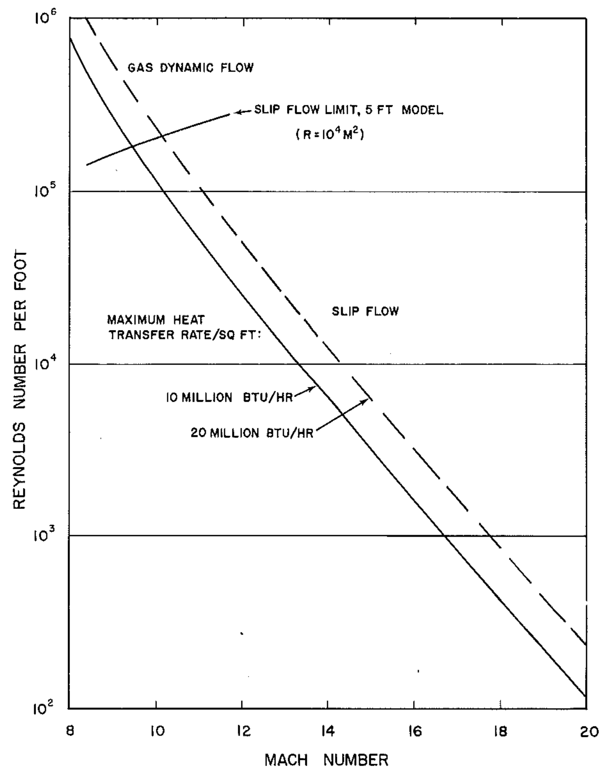

Fig. 7. Limiting Mach Number-Reynolds Number in Continuous Temperature-Simulating Tunnels, from Cooling Conditions

present. There may also be developed high-temperature materials capable of providing sufficient strength to contain the throat pressure at temperatures of several thousand degrees Rankine. But in fact, such improvements in cooling surface temperature become insignificant in comparison with the required air-supply temperature; and the heat transfer rates shown in Fig. 6 are several orders of magnitude higher than the present practical maximum, so that a very major engineering development must be postulated to make facilities of this type possible. Research workers in the field have realized this limitation, and have sought alternative approaches to obtain aerodynamic data at very high Mach numbers. Some of these approaches are described in the following three sections.

V. SEPARATION OF INDIVIDUAL PHENOMENA

In the preceding sections, the aim has been to provide a means of simulating all the parameters which enter into the aerodynamic behavior of air in high Mach number flight, including Mach number itself, viscosity, vibration, and dissociation. This is very evidently a difficult objective to attain, and some of the present approaches to the problem have confined themselves to the simpler problem of simulation of only a few of the parameters. As soon as this simplification is accepted, it is no longer necessary to use air as the working fluid, and other gases which permit the practical values of the critical parameters to be obtained more readily can be employed.

For example, a partial solution of the problem involving only the effects of Mach number and viscosity, and neglecting the variation in air properties due to vibration and dissociation, can be obtained by the use of a gas with a low liquefaction point such as helium. The work of Professor Bogdonoff of Princeton University (Ref. 10) is a good example of this approach. The low temperatures to which gaseous helium can be reduced in the test section permits rather high Reynolds numbers to be obtained, so that hypersonic viscous problems in the gas dynamic regime can be studied. The specific heat ratio γ is not correctly simulated, but it is evident that insistence on correct simulation would lead to a requirement for simulation of the variation which occurs in air in flight, and to the same practical problem as discussed in the last section.

An experimental approach to the effects of dissociation on the aerodynamic characteristics is also possible by change of the working fluid. This approach has been investigated at the Naval Ordnance Laboratory by Dr. Slawsky and his associates. They employ gases such as bromine and chlorine which dissociate at very low temperatures; this permits the phenomena associated with dissociation to be examined without serious cooling problems. At present the work in this area has been largely a matter of free-flight investigations using ballistic range techniques; the possibility of a wind tunnel employing readily-dissociating gases should also be explored.

VI. SHORT-DURATION HIGH MACH NUMBER FACILITIES

If air is to be used as the working fluid, it is evident that some means must be found to circumvent the cooling problem at the tunnel throat. One possibility is to reduce the time of operation of the high-temperature flow to such an extent that the heat transfer to the critical parts of the wall is not large enough to do damage. The limiting duration of a facility of this type is quite difficult to estimate because the extremely high heat-transfer rate, operating for a very short time, can produce two types of failure. It necessarily results in extreme differences in temperature between different portions of the tunnel walls, giving rise to thermal stresses which may be excessive; or alternatively, the maximum temperature on the inner wall may be sufficient to liquefy the surface at the points of maximum heat-transfer rate. A quantitative picture of this transient heating process can be obtained by application of classical heat-conduction theory. The temperature T at a distance x from the air surface in the metal after time t, is given by:

$$T(x,t) = \frac{H}{k}\left[2\sqrt{\frac{nt}{\pi}}\, e^{-\frac{x^2}{4nt}} - x\ erfc\left(\frac{x}{2\sqrt{nt}}\right)\right] \tag{4}$$

where $n = k/c\rho$

The time taken for the temperature at the surface ($x = 0$) to reach the liquefaction value T_m is then given by

$$t = \frac{\pi}{4}\, k\rho c\, \frac{T_m{}^2}{H^2} \tag{5}$$

If the wall is made of steel, insertion of numerical values leads to the rough result $t = \frac{10^{13}}{H^2}$, where H is measured in Btu per hour per square foot. However, steel is not the best material for this application; evidently the longest operating time is obtained with a material giving the maximum value of $k \rho c \; T_m^2$; this suggests the use of tungsten or a ceramic material. Tungsten, for example, gives twenty times the duration of steel, before its melting point is reached.

It appears at first sight that the condition of the inner surface of the wall reaching its melting point might be regarded as an upper limit to the operating time in intermittent facilities. There is however no structural problem presented by this condition; the high temperatures are confined to only a few thousandths of an inch in depth so that the majority of the wall still maintains its original strength. Furthermore, the quantity of wall material which is melted is not very large. In point of fact, some of the experimental facilities now in operation utilizing the shock tube principle, described in Ref. 11, already operate for a duration sufficient to produce melting and even evaporation. The main problem under these conditions is the pollution of the air, and eventually of the walls of the shock tube and tunnel, which makes it necessary to provide for frequent cleaning and replacement of critical areas.

We can assume, however, that the upper limit in operating time will bear some relation to the liquefaction time as obtained from Equation 5. This equation has therefore been employed, in conjunction with the maximum heat-flow rates from Fig. 6, to estimate the time required to liquefy the surface in an intermittent hypersonic facility. The results of this estimate, assuming a steel wall, are given in Fig. 8 (see page 27). It is evident from this figure that a reasonable Reynolds number can only be obtained if surface melting after a few milliseconds is accepted.

This result leads naturally to the use of a shock tube as the driving element of a hypersonic wind tunnel. The current position of development of "impulse" wind tunnels of this type has been summarized in Ref. 11. It is therefore sufficient in the present paper to point out some of the advantages of the shock tube as they relate to the problems discussed here:

1. The shock tube is probably the simplest equipment permitting high-pressure, high-temperature flows to be generated and terminated within a few milliseconds. It eliminates the practical problems of rapidly operating valves.

2. The shock tube considerably simplifies the problem of obtaining the high stagnation temperatures required in the operation of hypersonic temperature-simulating facilities. The temperature of the flow behind the shock can be many times greater than the temperature generated in the high-pressure driving chamber. It should be observed, however, that sufficiently high temperatures can certainly be obtained by the direct use of electric arc heating.

Two types of impulse tunnels utilizing shock tubes have been considered. In the reflecting type, the shock is driven along the tube and reflected from the wall at the further end. The flow behind the reflected shock is stationary and at extremely high temperature and pressure, so that it can be used as the air supply for a wind tunnel. There is, however, an alternative operation in which the shock is not reflected, but in which the end of the tube is expanded directly into the test area. This has the attraction that there is no sonic throat, and the throat heat-transfer problem then does not exist. In its place, the maximum heat-flow rate occurs at the wall behind the driving shock, which is subject to a high-pressure, high-temperature stream at a Mach number of about 2. The maximum heat-transfer rate under these conditions is about 60 percent of the throat value.

Taking into account this small alleviation of the wall heating problem with nonreflected operation of a shock-tube tunnel, and assuming an operating time of one millisecond before liquefaction, the limiting Mach number/Reynolds number relation for an impulse tunnel has been derived from the curves of Fig. 8, and is shown in Fig. 9. The assumed time of one millisecond is somewhat arbitrary; it is based upon the fact that commercial instruments for the measurement of transient pressures have a response time of the order of 10^{-4} to 10^{-5} seconds at best, and the total operating time should be at least an order of magnitude greater than the response time of the instruments. There is, of course, scope

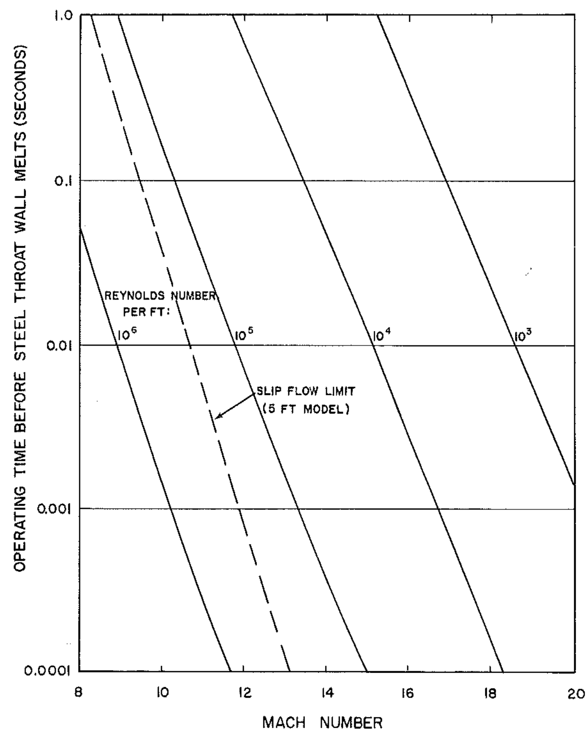

Fig. 8. Operating-Time Limitations in Intermittent Temperature-Simulating Tunnels Due to Overheating (Steel walls assumed; multiply by 20 for tungsten walls)

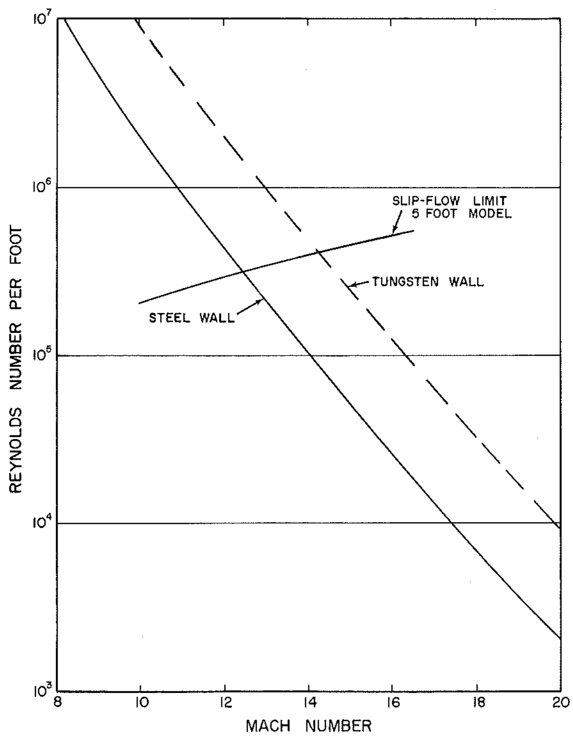

Fig. 9. Limiting Mach Number-Reynolds Number in Impulse Tunnels, as Determined by Wall-Melting; Operating Time One Millisecond

for future development of instruments with shorter response time; but it should be observed that a reduction in operating time by a factor of 10 will only increase the maximum Reynolds number by a factor of 10. Furthermore, the time of one millisecond corresponds to a distance of only twenty feet in flight at a Mach number of 20, so that further reduction in time presumably will produce instrumentation problems comparable with those of free-flight techniques.

Although these considerations may indicate that the curves of Fig. 9 are somewhat conservative, it is probable that the figure represents very approximately the upper limit at the present time. If the data are compared with corresponding data on Fig. 7 for the continuous temperature-simulating tunnel, it will be observed that the maximum Reynolds number has been multiplied by a factor of almost 100 by the use of shock-tube techniques.

At this point, it is appropriate to discuss the measurement of model temperature, which constitutes the chief problem in intermittent hypersonic tunnels. Since heating is likely to be the most important problem for the designer of high Mach number vehicles, it is essential to be able to measure rates of heat flow into the model in any high-speed test facility. It is obvious that this presents a difficult problem in impulse tunnels operating only for times of the order of a millisecond.

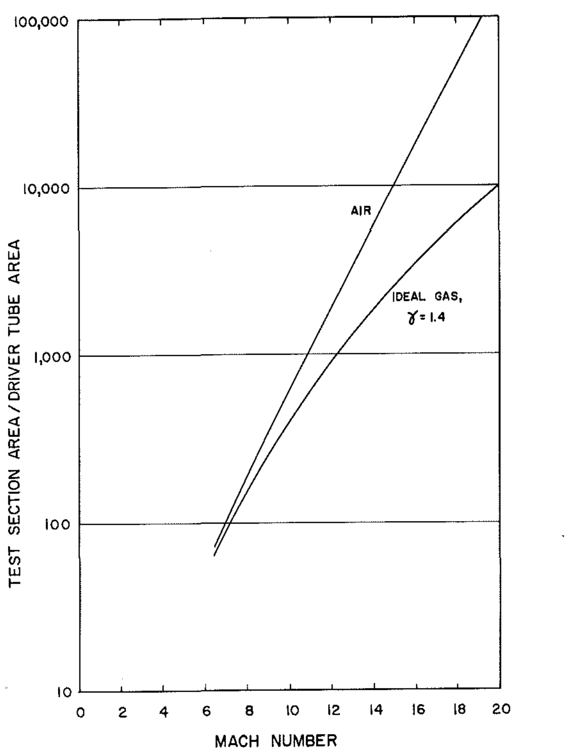

Fig. 10. Tunnel Area/Driver Tube Area Ratio in Nonreflecting Impulse Tunnels

It is useful to look at this problem quite generally, going back to the basic heat-transfer relation of Equation 2. A rough deduction from this equation is that the heat-transfer rate is inversely proportioned to the cross-sectional area of the air stream, so that the ratio of the heat transfer per unit area at the throat to the heat transfer per unit area on a test object (treated as a flat plate along the stream) is given roughly by the value of A/A^*. This ratio has already been given in Fig. 5. A similar relation holds between the heat transfer at the test section and the heat transfer behind the driving shock in a shock-tube tunnel; this is shown in Fig. 10.

These figures show that at Mach number of 19 or more in the shock tube or continuous tunnel (as noted earlier, the nonreflected shock-tube tunnel is a little better because of the absence of a sonic throat) the area ratio reaches a value of 100, 000. In other words, quite independently of the operating time, if the model and the driver wall are of similar construction, the rate of temperature rise at the model will be only 1/100, 000 of that at the worst point in the air channel. During the time taken for the hot spot to reach the melting point of steel, the model temperature rise would be only 0.02 $^{\circ}$F.

Of course, a typical model will not have merely flat-plate heat-transfer values. On the other hand, it will not have the very large heat-transfer rates which would correspond to an isentropic recompression to sonic speed. The case when the model is preceded by a normal shock has been considered, for comparison with the flat-plate condition. In this case, the worst heat-transfer rate is obtained when the air accelerates to sonic speed again behind the shock; at high Mach number, the value of ρv is a constant multiple,

$$\frac{(\gamma+1)}{(\gamma-1)}^{1/2} \frac{(\gamma+1)}{(2\gamma)}^{\frac{1}{\gamma-1}}$$

of the value in the ambient stream. The heat-transfer rate is then multiplied by a factor of only 1.66 with $\gamma = 1.4$.

To permit measurement of heat transfer with reasonable accuracy, it is necessary to obtain temperature rises of several degrees on the model. This can only be done by making the model surface so thin that the small amount of heat transfer during the run is enough to give significant temperature rise. It is estimated that, even with operating time sufficient to melt the driver wall in a shock-tube tunnel, the model wall thickness at the point where temperature is to be measured must be reduced to only a fraction of a micron. This implies that special techniques such as the evaporation of a very thin metal film on an insulated material must be adopted. These techniques are currently being developed, but it is not possible at this stage to predict their ultimate degree of success. The additional limit in Mach number and Reynolds number imposed by this requirement can therefore not be assessed at present.

VII. HEAT ADDITION TO A SUPERSONIC STREAM

The last few paragraphs have focussed attention upon a question which is really the basic problem of test facilities for the highest speed range. It is generally realized that the major problem of flight at such speeds is to avoid overheating of the vehicle walls. It has been determined (Section IV) that ground test facilities should be able to duplicate flight temperatures at Mach numbers above 12; so that at the same test-section density as in flight (which implies Reynolds numbers less than flight values if the model is smaller) the heating rate per unit area is just as large as in flight. But it is then inherent, in the facilities just discussed, that critical areas exist in which the heating rate is many thousand times the already large flight value.

In these circumstances, the value of ground test facilities to supplement flight test techniques is open to serious question. Ground facilities are only justified if they show appreciable gains in economy, convenience, and simplicity compared with flight tests; and the need to solve a grossly enlarged heating problem may offset the usual advantages of ground facilities in this respect.

There is one method of approach which appears to offer the possibility of of preventing most of the multiplication of the heating problem which occurs in the facilities discussed up to now. If the air stream could be first expanded to high supersonic Mach number at moderate temperatures, and then heated to the temperatures required for correct simulation of flight conditions, without reducing the Mach number sufficiently to approach sonic conditions, then the critical heating rates at the throat would be avoided, and the maximum rates would be of the same order as in flight. Measurable temperature rises on the test object would then not be accompanied by excessive heating elsewhere on the wind-tunnel walls.

It is not feasible to heat air traveling at high supersonic speeds by convective or conduction processes, but an electric arc of sufficient intensity across the supersonic stream can produce sufficiently high temperatures. The chief problem with such an arc is to prevent the ionized area from being swept downstream, blowing out the arc. In recent work by Smith and Early (Ref. 12) this blowout was avoided by a strong magnetic field; it is also possible to utilize a radio-frequency power supply for the arc, and thus to arrange very rapid re-ignition as a means of stabilizing the arc's position. At first sight, it appears that the ionization produced by the arc might be a deterrent to this type of facility. At the low temperatures of the supersonic stream, however, the ionization diminishes very rapidly. As a matter of fact, ionization of the air is less of a problem in this type of a wind tunnel than in the facilities discussed earlier. In these facilities at the highest Mach numbers there are regions of the stream in which the temperature exceeds $12,000^{\circ}\,R$, which is sufficient to give appreciable ionization and radiation. The additional losses from this cause, which have been ignored in the simplified approach described above, are of course avoided when the heating is confined to the area of high supersonic speed.

Very little work has been done up to the present on facilities of the supersonic-heating type; the best method of heat addition, and the resulting nozzle shape, need considerable experimental and theoretical work for their resolution. It is evident from the estimates presented here that this approach to high Mach number test facilities has far more promise than any of the alternatives at present being investigated.

REFERENCES

1. Wegener, P., et al. "NOL Hyperballistics Tunnel No. 4, Results I: Air Liquefaction." NAVORD Report 1742, January 1951.

2. Nagamatsu, H. T. "Galcit Hypersonic Research." Contribution to AGARD Conference, October 1954.

3. Allen, H. Julian. "Development of Two Hypersonic Test Facilities at the NACA Ames Aeronautical Laboratory." Contribution to AGARD Conference, October 1954.

4. Dodge, B. F. and Dunbar, A. K. "Investigation of the Co-existing Liquid and Vapor Phases of Solution of Oxygen and Nitrogen." Journal American Chemical Society, Vol. 49, pp. 591-610, 1927.

5. Applied Physics Lab., Johns Hopkins University. "Handbook of Supersonic Aerodynamics." NAVORD Report 1488, Vol. 5, August 1953.

6. Bethe, H. A. and Teller, E. "Deviations from Thermal Equilibrium in Shock Waves." Ballistic Research Laboratories Report No. X-117, (also published by Engineering Research Institute, University of Michigan).

7. Moore, L. L. "A Solution of the Laminar Boundary - Layer Equations for a Compressible Fluid with Variable Properties, Including Dissociation." Journal of the Aeronautical Sciences, Vol. 19, No. 8, pp. 505 - 518, 1952.

8. Crown, J. C. "The Laminar Boundary Layer at Hypersonic Speeds." NAVORD Report 2299, 1952.

9. Eckert, Ernst R. G. "Survey on Heat Transfer at High Speeds." WADC Technical Report 54-70, April 1954.

10. Bogdonoff, S. M., and Hammitt, A. G. "The Princeton Helium Hypersonic Tunnel and Preliminary Results above $M = 11$." Princeton Report R 260, July 1954 (preliminary copy of WADC Technical Report Number 54-124).

11. Dodge, J. A. "Ultra-High Temperature Aerodynamic Testing Facilities." AEDC-TN-54-61, April 1955.

12. Smith, H. L., and Early, H. C. "Investigation of Heating of Air Stream in a Wind Tunnel by Means of an Electrical Discharge." Engineering Research Institute, University of Michigan, Report No. 2154-3-F.